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Investigation of Propulsion System Requirements for Spartan Lite

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Abstract. This paper discusses the (chemical or electric) propulsion system requirements necessary to increase the Spartan Lite science mission lifetime to over a year. Spartan Lite is an extremely low-cost (<$10M) spacecraft bus being developed at the NASA Goddard Space Flight Center to accommodate sounding rocket class (40 W, 45 kg, 35cm dia by 1 m length) payloads. While Spartan Lite is compatible with expendable launch vehicles, most missions are expected to be tertiary payloads deployed by the Space Shuttle. To achieve a one year or longer mission life from typical Shuttle orbits, some form of propulsion system is required. Chemical propulsion systems (characterized by high thrust impulsive maneuvers) and electrical propulsion systems (characterized by low-thrust long duration maneuvers and the additional requirement for electrical power) are discussed. The performance of the Spartan Lite attitude control system in the presence of large disturbance torques is evaluated using the Treetops™ dynamic simulator. This paper discusses the performance goals and resource constraints for candidate Spartan Lite propulsion systems and uses them to specify quantitative requirements against which the systems are evaluated.

Background

Spartan Lite[1] is being developed at the NASA Goddard Space Flight Center (GSFC) as an extremely low-cost (<$10M) spacecraft bus to accommodate sounding rocket class (40 W, 45 kg, 35cm dia by 1 m length) payloads with science mission lifetime requirements of one year. While Spartan Lite is compatible with expendable launch vehicles, most missions are likely to be launched as tertiary payloads deployed on a Space Shuttle mission. Figure 1 depicts a typical Spartan Lite Spacecraft (without a propulsion system).

As a Space Shuttle tertiary payload, Spartan Lite cannot impose requirements on orbit insertion conditions; the standard Shuttle orbit altitude is 300 km. It is assumed that below 300 km the aerodynamic torque will degrade the system pointing performance below mission requirements. Therefore, this is regarded as the minimum useful altitude (end-of-life) for Spartan Lite.

Beginning with the launch of the first element of the International Space Station (ISS), 75% of all Shuttle missions visit the ISS. The ISS's altitude[2] is maintained such that its minimum altitude coincides with the arrival of visiting vehicles. The ISS's altitude at Shuttle arrival will vary from 350 to 425 km depending on solar flux. Even when deployed from the Shuttle during an ISS mission, Spartan Lite will decay to 300 km altitude in less than one year. To achieve a one year or longer mission life from typical Shuttle orbits, some form of propulsion system is required.

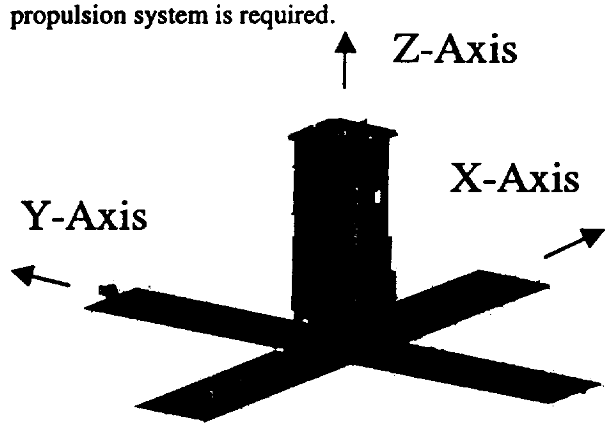

Figure 1. Deployed Spartan Lite Spacecraft

The terms defined below will be used throughout the remainder of this paper.

Deploy Altitude:	The altitude at which Spartan Lite is deployed from the Shuttle.
Initial Altitude:	The altitude to which Spartan Lite is boosted to begin its mission.
Mission Life:	Time to decay (due to drag) from the Initial Altitude to 300 km.

Modeling Assumptions

Solar Flux Prediction

Atmospheric density, and therefore orbit life, varies with solar flux. Figure 2 depicts three solar flux predictions, 95th, 50th, and 5th percentile (95th percentile represents a 95% probability that the solar flux will be at or below the predicted level) over 12.5 years, one solar cycle plus one year, beginning in January 1999. These curves are from the Marshall Space Flight Center monthly solar flux predictions, January 1998 release[3].

To assure a one year Mission Life under worst case conditions, 95th percentile solar flux is used in this paper. Because 95th and 5th percentile solar flux are similar at solar minimum, the 95th percentile case captures the full range of expected atmospheric conditions over the solar cycle.

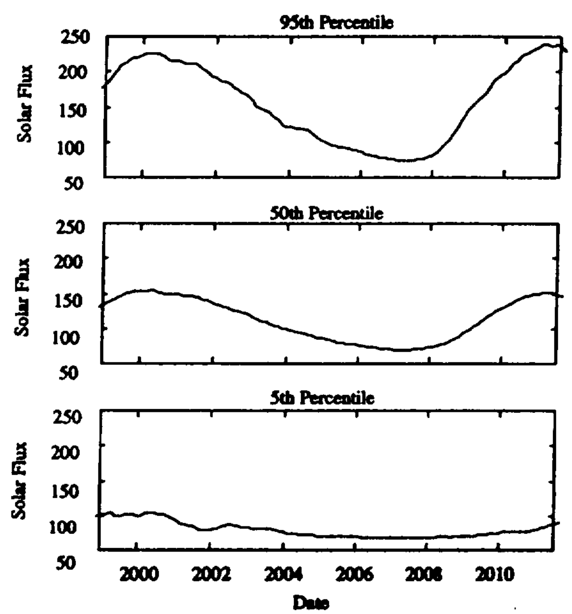

Figure 2. Predicted Solar Flux

Spacecraft Modeling

The Spartan Lite characteristics used to model atmospheric decay are summarized in Table I.

Table I. Characteristics for Drag Modeling

Empty Mass (no propellant)	146 kg
Average Projected Area	1.871 m^2
Drag Coefficient	2.2

Orbit Life Determination

Performance requirements for two mission scenarios are discussed in this section:

- Deploy at the standard Shuttle orbit (300 km altitude, 28.45° inclination) and boost to the Initial Altitude required for a one year Mission Life.
- Deploy from a Shuttle mission to the ISS (variable altitude, 51.6° inclination) and boost to the Initial Altitude required for one year Mission Life.

Figure 3 depicts the Initial Altitude required to achieve a one year Mission Life over an 11.5 year solar cycle from a standard Shuttle orbit at 28.45° inclination, assuming 95th percentile solar flux.

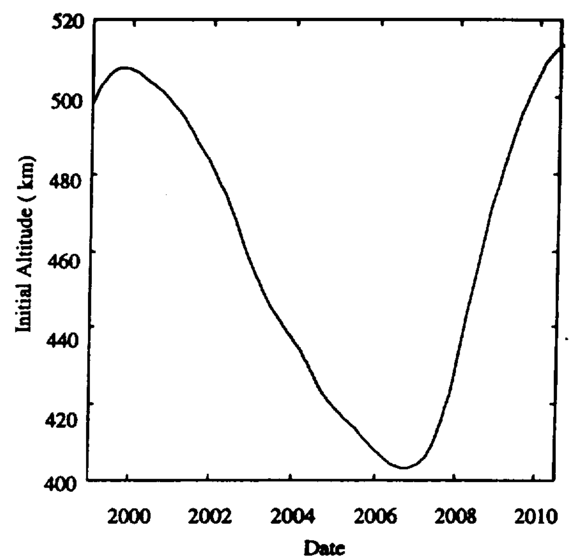

Figure 3. Initial Altitude for 28.45° Inclination

Figure 4 depicts the Initial Altitude required to achieve the same Mission Life for a 51.6° inclined orbit corresponding to a Shuttle/ISS mission. Because the orbit-averaged atmospheric density varies slightly with inclination, this altitude history is slightly lower than the 28.45° case.

Performance parameters and requirements to achieve the Initial Altitude from the Deploy Altitude are different for chemical and electric propulsion options. For that reason chemical and electric propulsion performance requirements are discussed separately in this paper. For purposes of this discussion, cold gas is considered a chemical propulsion technology.

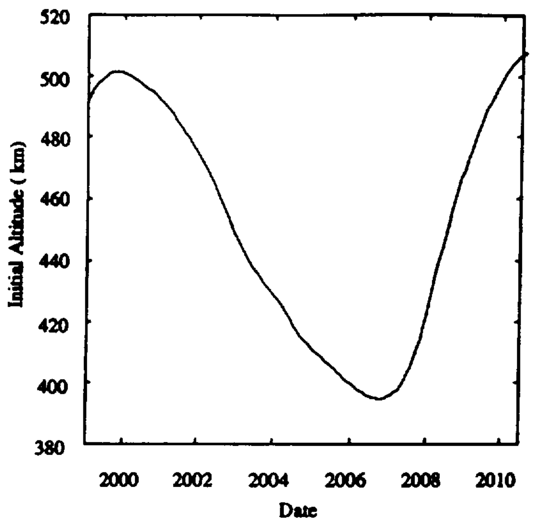

Figure 4. Initial Altitude for 51.6° Inclination

Chemical Propulsion Performance Requirements

Chemical propulsion systems have the following characteristics:

- High thrust, which permits modeling the propulsive maneuvers as impulsive events.
- All propulsive energy is stored in the propellant, so the system performance is not constrained by the available spacecraft power.

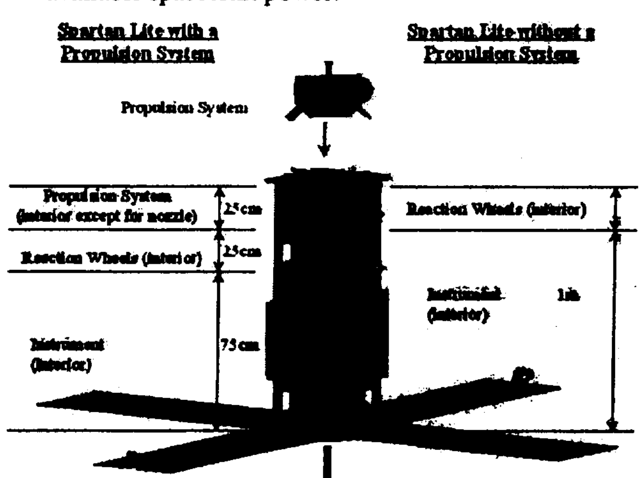

Figure 5. Volume Comparison with and without a Propulsion System

Propulsion System Mass/Volume Allocation

On Spartan Lite half the instrument mass and one fourth of the instrument length/volume can be allocated to the propulsion system. This permits a total propulsion system mass (empty mass plus propellant) of 22.7 kg packaged within a cylinder (excluding exhaust nozzle) of 35.5 cm diameter by 25 cm height. The thrust vector would be directed along the axis of symmetry of the cylinder, corresponding to the Z-axis of the spacecraft. Figure 5 compares the Spartan Lite baseline configuration with a configuration employing a propulsion system.

Performance Requirements

The ΔV to achieve the Initial Orbit for each mission scenario is addressed separately.

Standard Shuttle Orbit

For deployment from a standard Shuttle orbit, figure 6 depicts the following:

- A history of the Initial Altitude for a one year Mission Life over one solar cycle.
- The ΔV to boost from the 300 km Deploy Altitude to the corresponding Initial Altitudes.
- The percentage of the solar cycle for which the given ΔV is sufficient for a one year Mission Life. For example, a ΔV of 90 m/s achieves a one year mission for about 50% of the solar cycle.

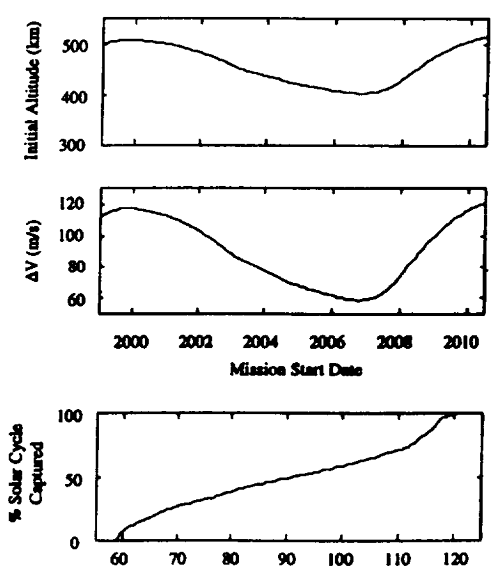

Figure 6. Altitude and ΔV Requirements when Deployed at the 300 km Standard Shuttle Orbit

A ΔV of 120 m/s is required to assure a one year Mission Life under worst conditions (solar maximum); 56 m/s is required under best conditions (solar minimum).

Shuttle/ISS Orbit

For deployment during a Shuttle/ISS mission, the ΔV is significantly less due to the higher Deploy Altitude and, to a lesser extent, the higher inclination of the ISS orbit. The ISS altitude is governed by several constraints but generally increases with solar flux over the solar cycle. Figure 7 depicts the following for Shutle/ISS mission and 95th percentile solar flux:

- A comparison of the Deploy (i.e., ISS) Altitude and Initial Altitude for a one year Mission Life over one solar cycle.
- The ΔV to boost from the ISS Deploy Altitude to this Initial Altitude.
- The percentage of the solar cycle the given ΔV is sufficient for a one year Mission Life.

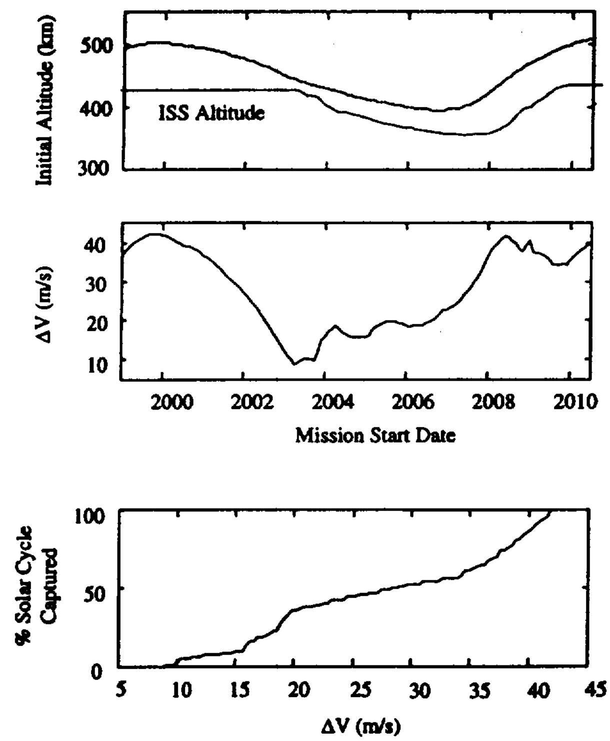

Figure 7. Altitude and ΔV Requirements when Deployed at the ISS Altitude

Figures 6 and 7 show that the maximum ΔV required to achieve a one year Mission Life from the ISS Deploy Altitude is less that the minimum ΔV from the standard Shuttle Deploy Altitude. Also, in Figure 7, the largest required ΔV occurs during solar maximum when the ISS altitude is no more than 425 km and Spartan Lite must achieve an Initial Altitude of 507 km.

Propellant Mass vs. Specific Impulse

Figure 8 shows the propellant mass required at a given Specific Impulse (I_s) to achieve a ΔV of 120 m/s (sufficient to assure a one year mission under all conditions).

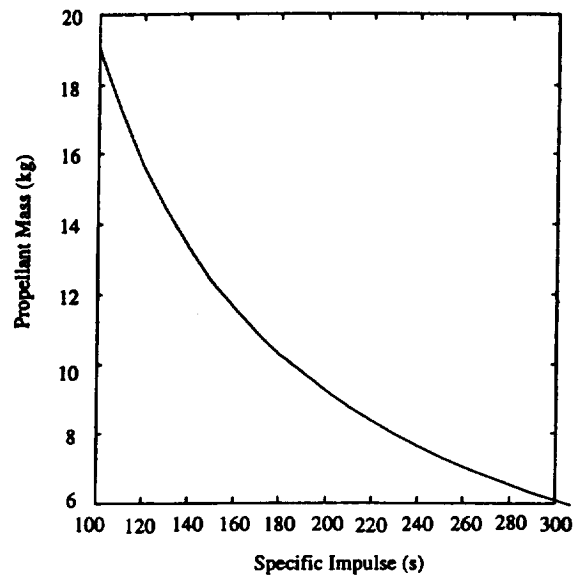

Figure 8. Propellant Mass vs Specific Impulse

Assuming 15.0 of the 22.7 kg propulsion system mass is allocated to propellant, an I_s of 125 s is sufficient to accomplish the mission. This is easily achieved by any contemporary chemical propulsion system except cold gas. A more limiting constraint is the volume allocation, which is addressed in later sections of this paper for each candidate engine or technology.

Attitude Control for Chemical Propulsion

The Spartan Lite Attitude Control System (ACS) sensors and actuators consist of an inertial three axis attitude sensor (star tracker or gyro), three reaction wheels and three magnetic torquers for momentum unloading. The reaction wheels and magnetic torquers are aligned with the body axes shown in Figure 1.

Misalignment of the thrust vector about the body X- and Y-axes results in a disturbance torque that must be absorbed by the reaction wheels. The magnitude of the torque can be computed using:

$$M = d \cdot F \cdot \sin(\theta) \qquad (1)$$

where M is the disturbance torque due to misalignment, d is the distance from the center of mass to the propulsion system (0.5 m), F is the thrust magnitude, and θ is the thrust vector misalignment angle measured from the thrust vector to the body Z-axis. The thrust vector misalignment angle was set conservatively at 0.5 degrees. The worst case wheel momentum and torque saturation occurs when all the misalignment is about a single axis. For this reason, it is assumed that all the misalignment was about the Y-axes for thruster sizing. The torque capacity of the wheels, M_{max}, limits the maximum allowable thrust, F_{max}, according to:

$$F_{max} = \frac{M_{max}}{d \cdot \sin(\theta)} \tag{2}$$

The momentum capacity of the wheels, H_{max}, limits the maximum thrust duration, t_{max}, according to:

$$t_{max} = \frac{H_{max}}{M_{max}} \tag{3}$$

The specifications for the wheels and magnetic torquers with associated maximum thrust and thrust duration are summarized in Table II.

A 31 N thrust over 60 seconds will produce a ΔV of approximately 10 m/s. To achieve the ΔV magnitudes shown in Figures 6 and 7, without saturating the reaction wheels, a sequence of orbit boost maneuvers is required. After each maneuver the momentum must be unloaded from the reaction wheels using the magnetic torquers before another boost maneuver can be performed.

Table II. Spartan Lite Reaction Wheel and Magnetic Torquer Specifications

Reaction Wheel	
Momentum Capacity	8.1 N·m·s
Torque Capacity	0.135 N·m
Magnetic Torquers	
Dipole Capacity	110 amp·m^2
Propulsion System	
Thrust Magnitude Limit	31 N
Thrust Duration Limit	60 s

A Treetops™ dynamic simulation was run to evaluate the Spartan Lite ACS performance during a 60 second orbit boost maneuver. Table III shows the parameters used in this simulation.

Figures 9(a) and 9(b) show that the Y-axis reaction wheel momentum and torque remain below the maximum values from Table II. Figure 9(c) shows that the maximum angular error from the desired pointing direction is small enough to produce only a slight deviation from the desired altitude after the boost maneuver. This error is acceptable because the orbit boost maneuver does not target a specific orbit. Its purpose is merely to raise the orbit.

Table III. Treetops™ Simulation Parameters

Spacecraft	
Moments of Inertia	I_{xx} = 32.1 kg·m^2 I_{yy} = 31.6 kg·m^2 I_{zz} = 22.6 kg·m^2
Products of Inertia	I_{xy} = -0.1 kg·m^2 I_{xz} = 0.9 kg·m^2 I_{yz} = 0.1 kg·m^2
PD Controller	
Sample Rate	10 Hertz
Delay	2 cycles (0.2 s)
Bandwidth	0.1 Hertz
Damping	0.8
Orbit Boost Thruster	
Thrust	31 N
Misalignment	0.5 degrees
Environmental Torques	
Aerodynamic	
Gravity Gradient	

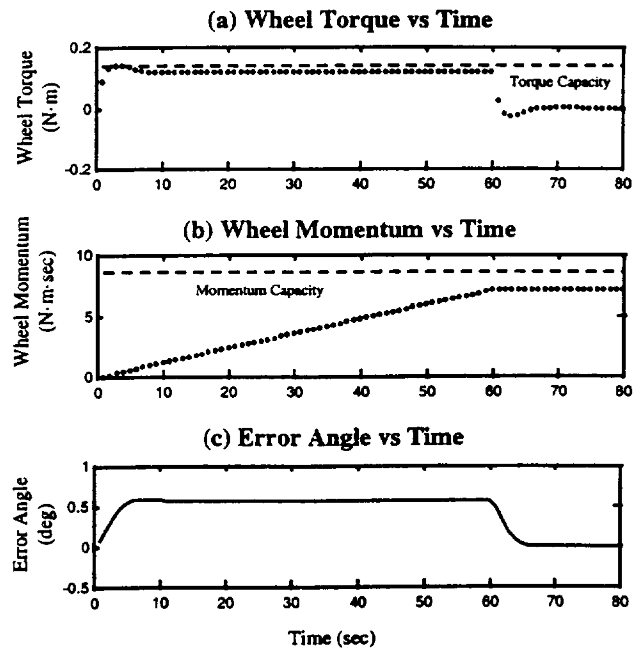

Figure 9. Reaction Wheel Torque, Momentum, and Spacecraft Angular Error During an Orbit Boost

Figure 10 shows the time the magnetic torquers require to remove angular momentum accumulated in the Y-axis reaction wheel for the altitudes of 300 and 500 km and an inclination of 28.45°. The results for 300 and 500 km orbits at an inclination of 51.6° are comparable. The periodic momentum build-up in Figure 10(a) is due to higher aerodynamic torque at the lower altitude.

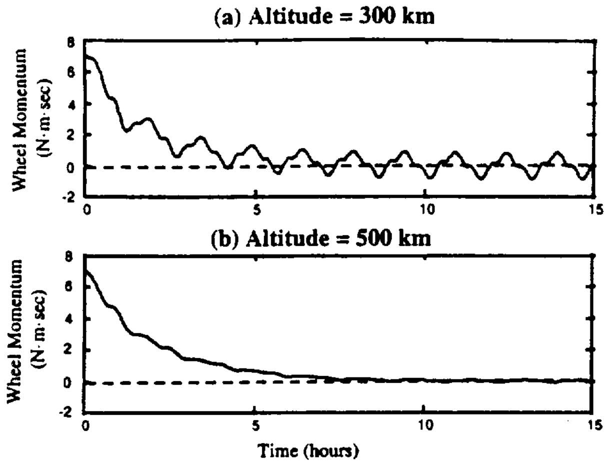

Figure 10. Time Required to Unload Momentum from Reaction Wheel

The result of this analysis is that anywhere from 1 to 12 maneuvers, separated by periods of about 5 hours where wheel momentum is unloaded, are required to achieve the ΔV magnitudes shown in Figures 6 and 7.

Electric Propulsion Performance Requirements

Electric propulsion systems have the following characteristics:

- Extremely low thrust, which requires modeling over several orbits in the presence of atmospheric drag.
- Thrust levels are constrained by the power available from the spacecraft.

Power Allocation

The Spartan Lite spacecraft employs body-fixed solar arrays. Since the spacecraft attitude for thrusting is referenced to the local vertical rather than solar inertial, the full solar array output is generally not available during thrusting periods. The electric thrust scenario, therefore, assumes the thruster operates for some fraction of the orbit (presumably corresponding to orbit night), then the spacecraft returns to Sun-point/solar inertial for the remainder of the orbit to recharge the batteries.

The operating power available to the electric propulsion system is constrained to 275 W by the maximum discharge rate of the batteries. An additional battery (assuming the propulsion system has mass/volume resources remaining) provides a good option to increase the maximum battery discharge rate and depth of discharge. This would permit a peak operating power of 400 W. This discussion uses a battery discharge rate of 275 W.

For a thruster operating for 40% of the orbit the total energy available is 72 W·hrs. The 275 W maximum discharge rate corresponds to an operating time of only 15.7 minutes. The remainder of this discussion uses an orbit average power of 48 W, which represents the 72 W·hrs averaged over the orbit.

The minimum thrust is determined by atmospheric drag at the Deploy Altitude and the time to climb to the Initial Altitude. The atmospheric drag is different for the two mission scenarios and is discussed in the subsequent sections. As a practical upper limit, the time to climb should be less than two months. Mass and volume constraints are the same as for chemical propulsion. Resource-driven constraints for electric propulsion are summarized in Table IV.

Table IV. Electric Propulsion System Constraints

Minimum Thrust	To overcome drag (varies)
Total Mass ($m_f + m_p$)	22.7 Kg
Volume	35.5cm dia x 25 cm ht
Operating Power	275 W peak 48 W orbit average
Time to Climb	Less than 2 months

Performance Requirements

A figure of merit for an electric propulsion technology can be specified by comparing the propulsion system performance parameters to the Spartan Lite operating constraints. The engine performance parameters of interest are the power efficiency, η, (propulsion power out divided by the electrical power in), and I_s. The Spartan Lite operating constraints of interest are the orbit-averaged minimum acceptable thrust, T_{min}, and the orbit-averaged maximum allowable electrical power, P_{max}. These parameters are related as follows:

$$\eta / I_s > (g/2) * (T_{min} / P_{max}) \quad (4)$$

In this relation g is acceleration due to gravity. Since P_{max} and T_{min} are orbit-averaged values, P_{max} in this relation is 48 W as discussed previously. Suitable values for T_{min} vary with mission scenario, as discussed in the following sections.

Standard Shuttle Orbit

Figure 11 depicts the following at a 300 km standard Shuttle orbit and 95th percentile solar flux:

- The Spartan Lite drag force at Deploy Altitude over one solar cycle.

- The percentage of the solar cycle the atmospheric drag is below a given value.

Note that the maximum drag at a 300 km Deploy Altitude is 5.6 mN.

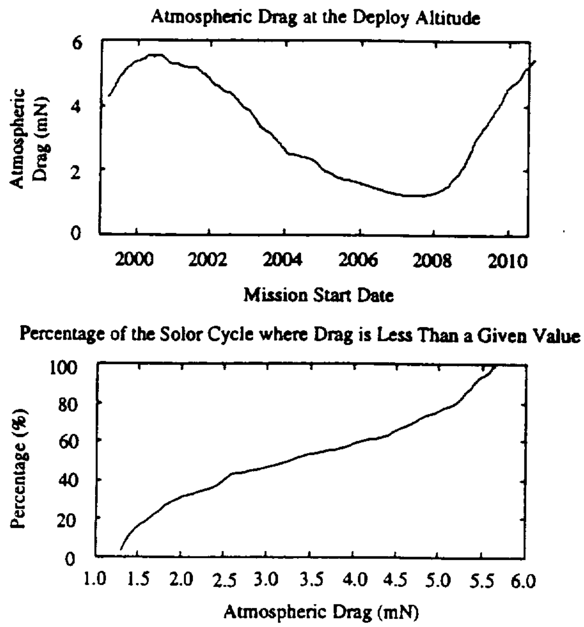

Figure 11. Atmospheric Drag at a 300 km Deploy Altitude and 28.45° Inclination

Figure 12 depicts the time and total ΔV, at solar maximum, required to reach the Initial Altitude for a one year Mission Life as a function of thrust level. Recall that for chemical propulsion 120 m/s is needed to reach this Initial Altitude.

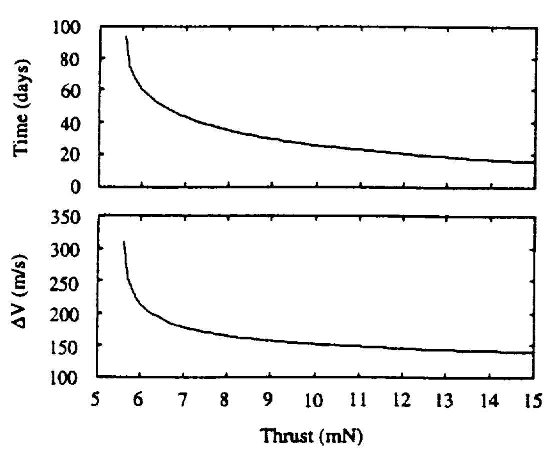

Figure 12. Time to Climb from 300 km to the Initial Altitude and the Resultant Total ΔV vs Thrust

To keep boost times and inefficiencies (due to thrusting against atmospheric drag) at reasonable levels, the minimum thrust should be at least 1.1 times the drag force at the Deploy Altitude (5.6mN x 1.1 = 6.16mN). Recall 6.16 mN is an orbit-averaged value, so the minimum thrust required from an electric propulsion system operated at a 40% duty cycle is 15.4 mN. At this level, the boost time is comfortably under two months and the total ΔV is less than 200 m/s.

To determine the figure of merit η/I_s, the orbit-averaged value of 6.16 mN is used for T_{min}. For the Spartan Lite spacecraft deployed from a standard Shuttle orbit, the figure of merit for a candidate electric propulsion system must satisfy:

$$\eta / I_s > 0.000629 \text{ s}^{-1} \qquad (5)$$

Shuttle/ISS Orbit

Figure 13 depicts the following when deployed at the ISS altitude with 95th percentile solar flux:

- The Spartan Lite drag force at the Deploy Altitude over one solar cycle.
- The percentage of the solar cycle the atmospheric drag is below a given value.

Note that the drag at the ISS Deploy Altitude at solar maximum, 0.68 mN, is almost a factor of 10 less than the drag at the standard Shuttle Deploy Altitude.

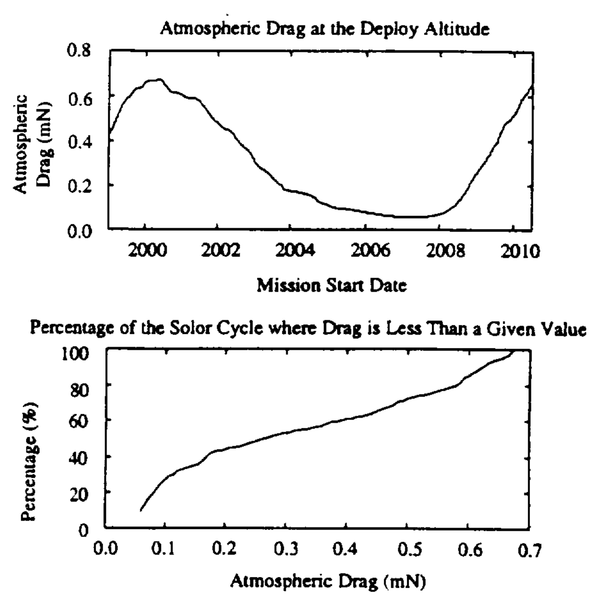

Figure 13. Atmospheric Drag at the ISS Deploy Altitude and 51.6° Inclination

Figure 14 depicts the time and total ΔV, at solar maximum, required to reach the Initial Altitude for a one year Mission Life as a function of thrust level. To keep the time to climb under two months, an orbit-averaged thrust of a least 1.8 mN is required. Therefore the minimum thrust required from an electric propulsion system at 40% duty cycle is 4.5 mN.

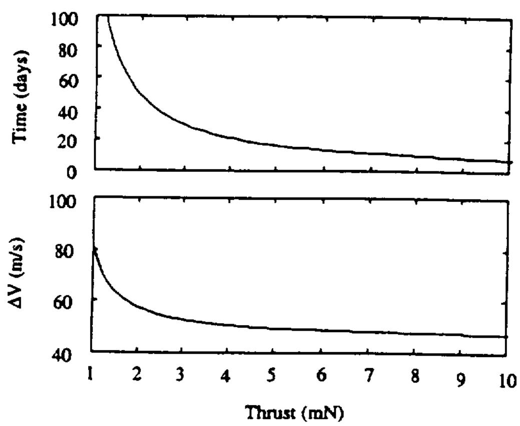

Figure 14. Time to Climb from the ISS Deploy Altitude to the Initial Altitude and the Resultant Total ΔV vs Thrust

As with the standard Shuttle orbit scenario, the orbit-averaged value (1.8 mN) is used for T_{min} to determine the figure of merit, η/I_s. When deployed during a Shuttle/ISS mission, the figure of merit for a candidate electric propulsion system must satisfy:

$$\eta/I_s > 0.000184\ s^{-1} \quad (6)$$

The time to achieve the Initial Orbit using a thrust of 4.5 mN at 40% duty cycle is 60 days, and the total ΔV is 58 m/s.

Chemical Propulsion Technology Survey

Below is a discussion of representative chemical propulsion technologies and their applicability to Spartan Lite within the requirements previously discussed.

Hydrazine Monopropellant

Hydrazine monopropellant is mature and has adequate performance (I_s between 180 and 220 s). It represents the performance benchmark against which alternative propulsion technologies are compared. Health, safety, and environmental concerns when using hydrazine result in a high recurring cost, particularly in Shuttle-launched spacecraft. While system costs are beyond the scope of this discussion, the objective of any propulsion system investigated is to reduce the recurring cost to a level significantly below that of hydrazine. This is presumed to be done by reducing the safety-imposed processing, shipping, and handling costs.

Compressed Cold Gas

Compressed cold gas is volume limited. A ΔV of about 8.4 m/s represents a reasonable maximum, permitting an altitude increase of a little better than 15 km. While this is never adequate to achieve a one year Mission Life for either of the scenarios discussed, such a system could provide useful mission extensions when deployed significantly higher than 300 km.

Modular Bipropellant System

A hypergolic bipropellant propulsion system developed for an existing program has potential as a low to moderate cost system for use on Spartan Lite. Adequate redundancy and inhibits are an intrinsic feature, and the system will be subjected to the Shuttle safety review process for its use on another Spartan mission. The recurring cost is mitigated by the modular architecture of the system, which is well matched to the Spartan Lite volume constraint. The I_s, approximately 275 s, is adequate, but the inherited system architecture limits propellant volume, permitting a total ΔV of 42 m/s when used on Spartan Lite. This is always adequate to achieve one year Mission Life on a Shuttle/ISS mission, but never adequate when deployed from a 300 km standard Shuttle orbit. The 310 N thrust level exceeds the Spartan Lite limit. However, the system is capable of pulsed operation at a 10% duty cycle to reduce the equivalent thrust level to under the Spartan Lite limit.

HydroxylAmmonium Nitrate (HAN)[4]

A HAN propellant is being developed as a non-toxic, environmentally benign replacement technology for hydrazine monopropellant. Current efforts under the direction of the NASA Lewis Research Center are working toward the demonstration of a flight-like 0.225 N "low-temperature" thruster with an I_s of 190 s. This is adequate for all one year Spartan Lite mission scenarios. The HAN propellant has a significantly higher density (1.4 times hydrazine), which is a good aid for packaging within the Spartan Lite volume constraint.

In the longer term, achieving higher I_s (in the range of 220 s) depends on the development of higher-temperature catalysts.

HAN appears to be a technology which has matured to the level for flight demonstration. The relatively inert, non-toxic nature of the propellant has the potential for significant reduction of recurring cost imposed by handling and safety-driven designs and analyses.

Electric Propulsion Technology Survey

Below is a discussion of representative electric propulsion technologies and their applicability to Spartan Lite within the requirements previously discussed.

Electric propulsion holds the potential for reduced recurring cost due to the use of non-toxic, inert fluids and materials for the propellant. For this reason technologies employing propellants such as mercury, ammonia, and hydrazine are not addressed.

Pulsed Plasma Thrusters (PPTs)[5]

The combination of the high I_s (1000 s or more) and low power efficiency (0.10) of PPTs yields an η/I_s of 0.0001 s^{-1} or less. This is well below what is necessary for an electric thruster for either Spartan Lite mission scenario.

Hall Thrusters[6,7]

Compared to PPTs, Hall thrusters have a higher operating efficiency (0.28- 0.35) and similar I_s (1000 s), yielding an η/I_s of around 0.00028 - 0.00035 s^{-1}. This is adequate for a Spartan Lite spacecraft deployed during a Shuttle/ISS mission, but not adequate when deployed from the 300 km standard Shuttle orbit. Hall thrusters employing xenon as propellant are available at power levels close to the Spartan Lite 275 W limit.

Water ResistoJets[8,9]

Water ResistoJets claim good power efficiency (60%) and relatively low I_s (200 s) for an η/I_s of 0.003 s^{-1}, adequate for either mission scenario. Operating at the allowable orbit-averaged power of 48 W, a Water ResistoJet could provide an orbit-averaged thrust of 0.03 N, a level at which thrust losses due to atmospheric drag become small to negligible. Packaging may be a problem due to the volume of water required.

Recent development of Water ResistoJets has been sparse; earlier development concentrated on higher thrust and power levels suitable for Space Station reboost.

Microwave Electric Thruster (MET)

METs using water as a propellant are claimed to be able to provide an I_s of 450 s at an efficiency of 0.28, which produces an η/I_s of 0.000622 s^{-1}. This is very close the 300 km deploy scenario requirement (0.000629 s^{-1}), which suggests that the MET is a good match to the Spartan Lite requirements. However, predicted MET thruster performance has yet to be demonstrated.

Summary

A number of chemical propulsion technologies were examined. The HAN propellant thruster appears to be the most viable candidate among all propulsion technologies surveyed for Spartan Lite. It promises adequate performance with lower recurring costs than hydrazine and has been developed to a level adequate to justify a flight demonstration. Another option, which provides adequate performance only when the Spartan Lite spacecraft is deployed during a Shuttle/ISS Mission, is an existing modular propulsion system using bipropellants.

Spartan Lite imposes a set of operating constraints (low available power, minimum thrust to overcome drag) different from typical electric thruster applications. The Spartan Lite performance requirements favor electric propulsion options with lower specific impulse and high operating efficiency. Among the technologies surveyed, none are available with adequate performance for all mission scenarios and development maturity to employ on Spartan Lite in the near term. Hall thrusters provide adequate performance for the Spartan Lite spacecraft when deployed during a Shuttle/ISS Mission, but not when deployed from a 300 km standard Shuttle orbit.

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Biographies

Urban, Michael G.

Mike Urban received a B.S. in Aerospace Engineering from the University of Cincinnati in 1978, an M.S. in Mechanical Engineering (Space Systems) from The George Washington University in 1994, and is currently pursuing an MS in Electrical Engineering from George Mason University. Mr. Urban has been employed at GSFC for seven years. He is currently the Sructures and Mechanical Systems Lead for the Polar Operational Environmental Satellites (POES). Prior to that assignment, Mr. Urban spent six years working most aspects of Hubble Space Telescope Operations and Servicing, and one year working various Spartan Lite and Spartan 400 spacecraft planning and mission design issues. Prior to working at GSFC, Mr. Urban was employed for thirteen years at the Caltech Jet Propulsion Laboratory (JPL); ten years in Voyager Spaceflight Operations and Flight Software Systems Engineering and three years at JPL's Space Station Freedom office in Reston, VA. While an undergraduate student, Mr. Urban worked at the Johnson Space Center in Mission Operations as a Co-Operative Education student.

Gruner, Timothy D.

Tim Gruner received his B.S. in Electrical Engineering from the University of Delaware has been working at GSFC for the past thirteen years. His present position is as a System Engineer for the Spartan Lite advanced carrier. Previous responsibilities at GSFC have included Attitude Control Electronics Engineer for the Solar Anomalous and Magnetospheric Particle Explorer (SAMPEX) mission, Lead Mission Unique Electronics Engineer for the Fast Auroral Snapshot Explorer (FAST) mission, and Lead Instrument Engineer on the Cosmic Ray Upset Experiment (CRUX).

Morrissey, James R.

Jim Morrissey received a B.S. in Chemistry from McGill University in 1990. After taking prerequisite Physics and Mechanical Engineering at the University of Massachusetts, he received an M.S. in Astronautics from The George Washington University in 1995. Mr. Morrissey is currently employed as a Co-Operative Education student in the Flight Dynamics Analysis Branch at GSFC while pursuing a doctorate in Spacecraft Control Systems at the University of Maryland. In the past four years Mr. Morrissey has worked on attitude control systems for the Spartan series of satellites, including Spartan 201, 204, 206, 207, Spartan Lite and Spartan 251.

Sneiderman, Gary

Gary Sneiderman received a B.S. in Mechanical Engineering from Tulane University in 1988 and a M.S. in Mechanical Engineering (Space Systems) from The George Washington University in 1997. In his ten years at GSFC, Mr. Sneiderman has worked as the lead mechanical engineer on several small satellites, including the Submillimeter Wave Astronomy Satellite (SWAS) and the Spartan advanced carriers. He is currently the lead mechanical engineer for the Spartan Lite and Spartan 400 carriers as well as supporting the new business activities of the Spartan Project.